BELLCOMM, INC.
955 L'ENFANT PLAZA NORTH, S.W. WASHINGTON, D. C. 20024

SUBJECT: Orbit Selection and Propellant Requirements for a "Minimum Apollo Applications Program" - Case 610

DATE: December 20, 1968

FROM: W. L. Austin

ABSTRACT

Trajectory analysis confirms the basic feasibility of a "Minimum Apollo Applications Program." However, an elliptical orbit must be used to stay within the Apollo Block II service module RCS propellant limits. An orbit is selected for the Mission Support Module Laboratory (MSML) having an altitude of 130 x 240 nm and an inclination of 29°. This provides an orbit lifetime of at least 60 days, an RCS propellant requirement for rendezvous maneuvers and backup deorbit that is within the Block II tank capacity, an ample payload margin for the MSML launch and the CM/SM launch, and reasonable launch window freedom to accomplish rendezvous.

(NASA-CR-103695) ORBIT SELECTION AND PROPELLANT REQUIREMENTS FOR A ''MINIMUM APOLLO APPLICATIONS PROGRAM'' (Bellcomm, Inc.) 23 p

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MEMORANDUM FOR FILE

1.0 Introduction

The material presented in this memorandum was prepared in support of a study of a "Minimum Apollo Applications Program" described in Reference 1. Our task was to determine suitable orbit parameters for the Mission Support Module Laboratory (MSML) of the study and a scheme for rendezvous of a CM/SM with the MSML subject to the following constraints:

(1) The MSML must have a 29-degree inclination and a 60-day lifetime when a $+2\sigma$ atmospheric density is assumed.

(2) The SM-RCS propellant requirements must not exceed the Apollo Block II total of 1278 pounds.

(3) Both launch vehicles are Saturn IB's.

(4) There must be at least one daily launch opportunity for the CM/SM over a 5-day period after the MSML launch.

The requirement that the SM-RCS consumption not exceed the Apollo tank capacity imposes restrictions on both the rendezvous and backup deorbit maneuvers, which are the two principal demands on the RCS. Previous studies have shown that the hybrid-stable-orbit rendezvous is about the most efficient in terms of RCS propellant consumption, but even if this technique were used, some 600 pounds will be required for the rendezvous phase of the mission, leaving about 600 pounds for backup deorbit. This suggests flying the mission in an elliptical orbit with backup deorbit executed at apogee only. However, this approach is limited for, as the eccentricity is increased (and the orbit parameters are adjusted to hold lifetime constant), the deorbit requirements of the prime system, the SPS, increase, because the SPS must be capable of safely deorbiting the CM/SM from any point in the orbit. In fact the SPS deorbit Δv from perigee (worst case) increases faster than the RCS backup deorbit Δv from apogee decreases. Thus, as the RCS backup deorbit requirement is reduced, the total CM/SM weight increases.

The increased SPS propellant can easily be accommodated in the available tanks which are very large, but the net increase in CM/SM weight poses a problem, as it can exceed the launch vehicle capability if the orbit eccentricity is increased too far. The requirement for a CM/SM launch opportunity on each of five days after MSML launch further modifies the allowable orbit eccentricity since CM/SM insertion by a direct ascent launch into points on the orbit other than perigee (the optimum) will be necessary.

The following sections present details of the analysis and selection of orbit parameters including calculation of orbit lifetime and deorbit requirements, an SM propellant budget, launch vehicle performance and launch window analysis. It should be noted that this is a preliminary analysis undertaken solely to demonstrate the feasibility of the MSML mission. Hence, trajectory optimization was not carried beyond the point necessary to accomplish the desired objective.

2.0 Lifetime and Deorbit Requirements

An orbit lifetime analysis of the docked CM/SM-MSML configuration was performed to determine which orbits would have a lifetime of sixty days (Reference 2). A $+2\sigma$ atmospheric density was used in the calculations, ensuring with high probability that the actual lifetime would exceed the calculated value. A circular orbit established at 170 miles decays in 60 days. The same lifetime can be had for an elliptical orbit with a lower perigee and higher apogee. Figure 1 shows the locus of all such combinations including five sample cases that were analyzed further and also appear in Table I.

Computation of the SM-RCS backup deorbit requirements for each of the five cases requires determination of the SPS propellant requirements to obtain the proper CM/SM initial weight. We have chosen to use the SPS only for the primary deorbit maneuver; this permits sealing the system after fueling and possibly increases reliability. It is possible because, as will be shown, the RCS propulsion is sufficient for the necessary rendezvous and backup deorbit maneuvers. The hybrid stable orbit technique is used for rendezvous. This requires several small burns of less than 20 ft/sec impulse, too small for efficient use of the SPS engine.

The reentry trajectory is usually specified in terms of acceptable combinations of velocity and flight path angle at 400,000 feet (Reference 3). However, for preliminary planning an acceptable approximation for low-altitude AAP missions is to target the SPS deorbit trajectory for a vacuum perigee of -40 nm and the RCS backup deorbit trajectory for a vacuum perigee of +30 nm. This method was used to calculate the deorbit requirements detailed below.

Deorbit from perigee is the most demanding case for the SPS and will therefore determine the SPS requirement for the mission. The maximum Δv requirements for the prime deorbit system, the SPS, have been tabulated in Table 1 for the five cases considered. The corresponding SPS propellant quantities, assuming a CM/SM weight of 25,442 lbs (without SPS propellant), are also given. Since the SPS is used only for deorbit, the entire SPS budget is based on this requirement.

The RCS backup deorbit requirements were calculated for an apogee burn that reduces perigee height from the mission orbit value to +30 nm. It was assumed that the backup system must deorbit the CM/SM with the full SPS propellant load still aboard. Thus the corresponding SPS propellant weight was included in the CM/SM overall weight before the RCS calculations were made.

Table II presents resulting RCS budgets and the margin for each case with respect to the 1278 lb Block II RCS usable propellant limit. Cases 1 and 2 have negative propellant margins; whereas Cases 3, 4, and 5 have progressively increasing positive margins. The Case 4 orbit, 130 x 240 nm, was somewhat arbitrarily chosen for the MSML simply because it is the most circular orbit where the propellant margin is greater than 50 lbs.

3.0 CM/SM Propellant Budget

The hybrid stable orbit rendezvous technique was chosen for rendezvous because it requires relatively little RCS propellant. Figure 2a illustrates the relative motion of the CM/SM in a coordinate system centered at and moving with the MSML. This is a curvilinear coordinate system; the vertical axis denotes altitude difference, the horizontal axis denotes true anomaly difference multiplied by a, the MSML orbit semi-major axis, with the MSML's forward direction to the left.

The nominal rendezvous sequence is as follows:

(1) The CM/SM is inserted by the launch vehicle at Point 1, about 60 nm behind the MSML with zero relative velocity.

(2) The CM/SM coasts at this stable point for a suitable time.

(3) A retrograde burn of the RCS at Point 2 creates a perigee height differential of such magnitude that the next apogee, Point 3, will occur 20 nm behind the MSML.

(4) The terminal phase is initiated from Point 4 by an RCS burn, called TPI, which is calculated to eliminate the position separation exactly one orbit later. This burn will be posigrade if the initial separation between the MSML and CM/SM is greater than 40 nm.

(5) The terminal phase of rendezvous is completed at Point 5 with a braking burn, TPF, which eliminates the relative velocity.

Insertion of the CM/SM will be made as close as possible to perigee of the MSML for reasons of launch efficiency, but within the launch window limitations explained in Section 4.0. Position and velocity components at insertion will be in error due to guidance inaccuracy and Saturn IB performance deviations, with the largest dispersion expected in downrange position. The 60 nm offset in the insertion position is selected specifically to accommodate 3σ dispersions of ± 40 nm in this coordinate without risking insertion forward of the 20 nm point.

The Δv and propellant required in step 3 to initiate the phasing maneuver is proportional to the distance to be made up. Allowing as large a range as 40 ± 40 nm incurs a heavy cost. This cost can be halved if two orbits are used for phasing instead of one, as illustrated in Figure 2b. Accordingly, we assume 2 phasing orbits would be used if the insertion offset turns out to be greater than 60 nm and 1 orbit if it is less. The coast period (step 2) would be adjusted to keep the total rendezvous time the same. In this way, the phasing maneuver can be held to less than 15 ft/sec. The coast period may also be adjusted to improve ground coverage of critical portions of the flight and to control the time of final approach, and thereby, the solar illumination of the target vehicle.

The final approach transfer path has been assumed to occupy a full 360° of travel, like the phasing maneuver, in order to do it with the least possible fuel. It remains to be seen whether this is acceptable. Closing speed and line of sight rate will have to be tested with man-in-the-loop simulations. Normally, a 270° transfer would be used, but the substantially larger Δv at both TPI and TPF make this unattractive in this minimum performance study.

A detailed SM-RCS propellant budget is presented in Table III and a timeline is presented in Table IV. Sufficient propellant is budgeted for the phasing maneuver for a maximum required catchup rate of 40 nm per orbit. For all maneuvers excluding backup deorbit, the total SM-RCS propellant required is 589 lbs. With backup deorbit and the gaging allowance included, the total SM-RCS propellant required is 1197 lbs. Hence the use of the hybrid-stable-orbit-rendezvous technique with the MSML in a 130 x 240 nm orbit results in a usable SM-RCS propellant margin of 81 lbs.

4.0 Launch Vehicle Performance and Launch Window Analysis

The MSML orbit is 130 x 240 nm at an inclination of 29.0 degrees. The rendezvous technique chosen is the hybrid-stable orbit, which requires the CM/SM to be inserted into the same orbit as the MSML. Thus the insertion altitudes and true anomalies of the CM/SM could range from 130 to 240 nm, and 0 to 360 degrees respectively.

The ability of the Saturn IB launch vehicle to insert the unmanned MSML into the selected orbit, 130 x 240 nm, was investigated using the BCMASP simulator (Reference 4). Runs were made both with and without SLA/Nose Cone jettison during boost. The results plotted in Figures 3 and 4 show that the MSML payload can be inserted at any point in the desired orbit regardless of whether SLA/Nose Cone jettison is used. Figures 3 and 4 are asymmetrical because when the insertion true anomaly is negative, the targeted flight path angle is negative. This is a more difficult condition to achieve. This will also be true for the CM/SM.

The CM/SM launch vehicle capability vs true anomaly is presented in Figure 5. For insertion true anomalies near apogee from -140 to +145 degrees, the CM/SM launch vehicle cannot insert the required minimum payload into orbit by a direct launch. This is not a serious limitation, however, because an orbit can be established at MSML launch time that permits CM/SM insertion near enough to perigee at each launch opportunity over the next 5 days.

It should be noted that a multi-burn ascent with a parking orbit and Hohmann transfer would have increased the payload significantly and made it insensitive to the relative position of the launch pad and perigee. However, the SPS engine would have to be used, conflicting with our desire to keep this system sealed until deorbit.

The launch window is defined as the time interval during which the plane of the target orbit is within the plane-change capability of the CM/SM launch vehicle. Launch opportunities are the times within the windows that the phasing is correct for launch. It is beyond the scope of this preliminary analysis to determine when the launch opportunities occur in the launch windows. However, we are concerned with whether or not an opportunity occurs for every window.

The launch period of the CM/SM considered is 5 days, including the day of MSML launch. Using the assumptions of a CM/SM launch-vehicle plane-change capability of 0.5 degrees, a CM/SM launch-site latitude of 28.522 degrees, an MSML instantaneous launch azimuth of 82.6 degrees, and the MSML orbit parameters as inputs to the "Launch Window Determination Program" (Reference 5), results in launch window lengths of 108.37 minutes for Day 0 and 127.67 minutes for Days 1 thru 5.

The period of the MSML orbit relative to a rotating earth is 96.89 minutes. Since this period is less than both CM/SM launch window sizes, at least one CM/SM launch opportunity will occur in every launch window. Thus, over the 5-day period there is a minimum of six CM/SM launch opportunities. Daily times of the launch window opening and closing, were generated by the "Launch Window Determination Program," and are presented in columns 2 and 3 of Table V. Time 0 is MSML launch.

With the launch windows for the CM/SM established, the final parameter which must be determined is the true anomaly of the MSML insertion which will provide acceptable values of CM/SM insertion anomaly over the 5-day launch period. The variation in true anomaly at CM/SM insertion is due primarily to the daily rotation of the line-of-apsides, and the motion of the CM/SM launch site relative to the plane of the MSML orbit as the CM/SM launch site passes through the launch window. The rotation of the line-of-apsides causes a change in the position of the MSML subperigee point relative to the opening of the launch window.

If the CM/SM is launched simultaneously with the MSML, the targeted true anomaly of the CM/SM, W_C, is equal to the true anomaly, W_T, of the MSML at MSML insertion. W_C for the remainder of Day 0 is approximated by:

$$W_C \simeq W_T + \left[t_2(\Omega_e \cos\Phi_L - \dot{\Omega}_1 \cos\Phi_L + \dot{\omega})\right] \qquad (4.1)$$

where: t_2 = the time in the launch window, 0 to 108.37 minutes
Ω_e = 0.25068 deg/min (rotation rate of the earth)
$\dot{\Omega}_1$ = -0.00504 deg/min (nodal precession rate)
$\dot{\omega}$ = 0.00814 deg/min (rotation rate of the line-of-apsides)
Φ_L = CM/SM launch site latitude.

For succeeding days, Days 1 thru 5, W_C is approximated by:

$$W_C \simeq W_T + \left[-4.52 - \dot{\omega}\, t(K) + t_3(\Omega_e \cos\Phi_L - \dot{\Omega}_1 \cos\Phi_L + \dot{\omega})\right] \qquad (4.2)$$

where: K = 1, 2, 3, 4, 5
$t(K)$ = the time of launch window opening on the Kth day
t_3 = the time in the launch window, 0-127.67 minutes.

Equations 4.1 and 4.2 assume the eastward motion of the launch site due to earth rotation is substantially parallel to the orbit plane. Since the latitude is nearly equal to the inclination and the launch site remains within $\pm .5$ degrees of the plane over the period of interest, this is a pretty good approximation.

Equations 4.1 and 4.2 differ because on Day 0 the launch window begins with the MSML launch which is equivalent to the first coplanar crossing on succeeding days. See Figure 6. The second term in Equation 4.2 is the central angle, measured in the plane of the MSML orbit, from the opening of the launch window to the first coplanar crossing. The third term accounts for the change in central angle of the subperigee point between launch window openings.

In Table V specific launch window limits are listed for each day along with the range of insertion anomaly that would be required of the CM/SM. If the MSML is inserted at perigee, $W_T = 0$, the CM/SM insertion varies from 26.76° at the close of Day 0 window to -61.81° at the opening of the Day 5 window. As can be seen from Figure 5, this range is entirely acceptable. Additional payload margin can be assured, however, by inserting the MSML at $W_T = +18°$. As shown in the final column of Table V, the CM/SM insertion would remain within $\pm 44°$ true anomaly.

5.0 Summary

An MSML orbit and CM/SM rendezvous technique were found which met the constraints set forth in the "Minimum AAP" study. The MSML orbit selected is 130 x 240 nm, which has an SM-RCS backup deorbit requirement of 540 lbs. Utilizing the hybrid-stable-orbit-rendezvous technique, the other SM-RCS propellant requirements are 589 lbs. The total SM-RCS propellant requirements, including gaging are 1197 lbs which results in a positive margin of 81 lbs.

At least one CM/SM launch opportunity per day over the 5-day launch period is guaranteed by inserting the MSML at a true anomaly of +18 degrees. Hence, within the prescribed constraints the basic feasibility of the MSML mission has been established.

6.0 Acknowledgements

The author is indebted to Mr. R. C. Purkey for performing the lifetime analysis and determining the backup Δv requirements. The author is also indebted to Mr. P. H. Whipple and Miss J. C. Gurasich for determining the payload capabilities of the MSML and CM/SM launch vehicles.

W. L. Austin

1025-WLA-dcs W. L. Austin

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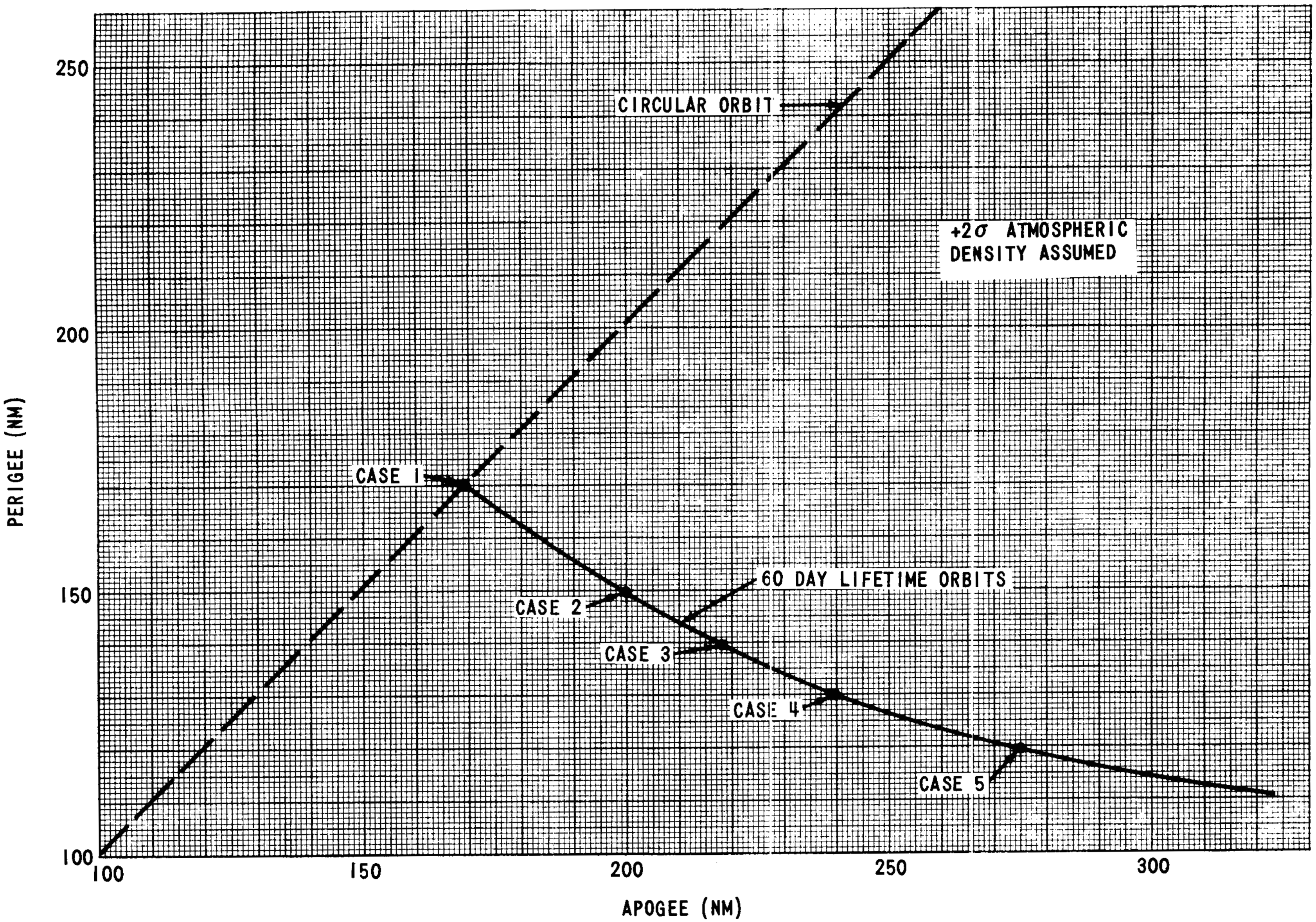


FIGURE 1 - GUARANTEED 60-DAY LIFETIME ORBITS

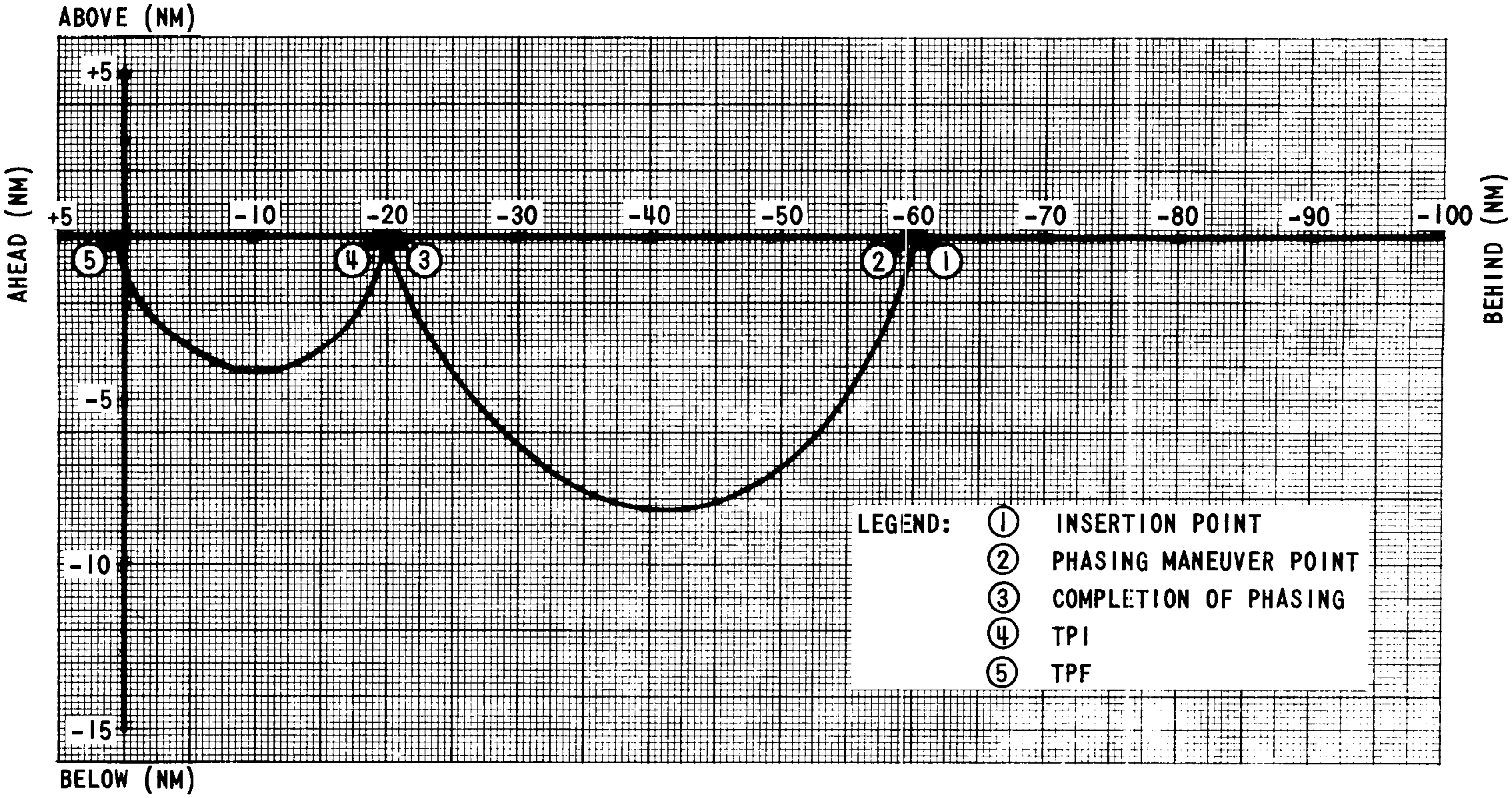


FIGURE 2a - CM/SM - MSML RELATIVE MOTION PLOT (NOMINAL CM/SM INSERTION)

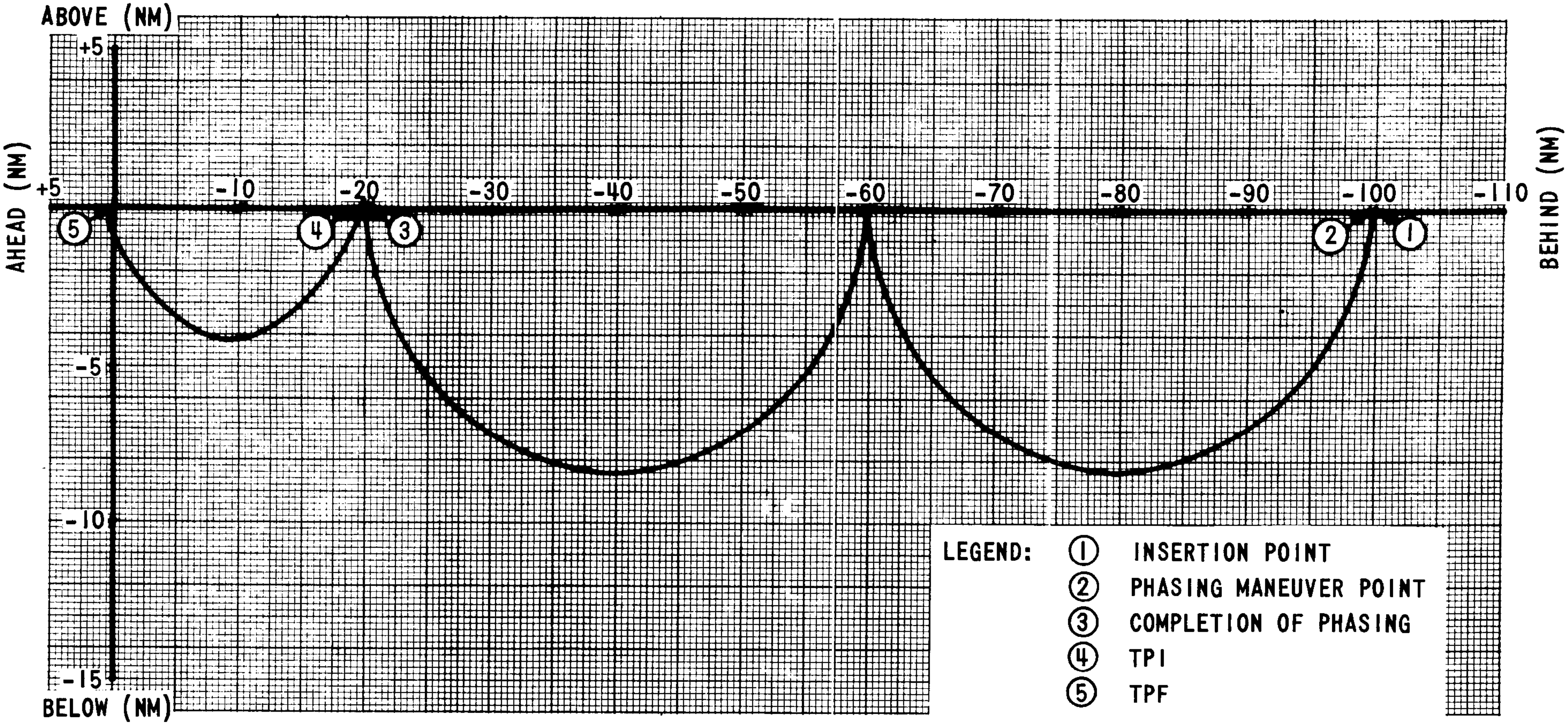


FIGURE 2b - CM/SM - MSML RELATIVE MOTION PLOT PLOT (-3σ CM/SM INSERTION)

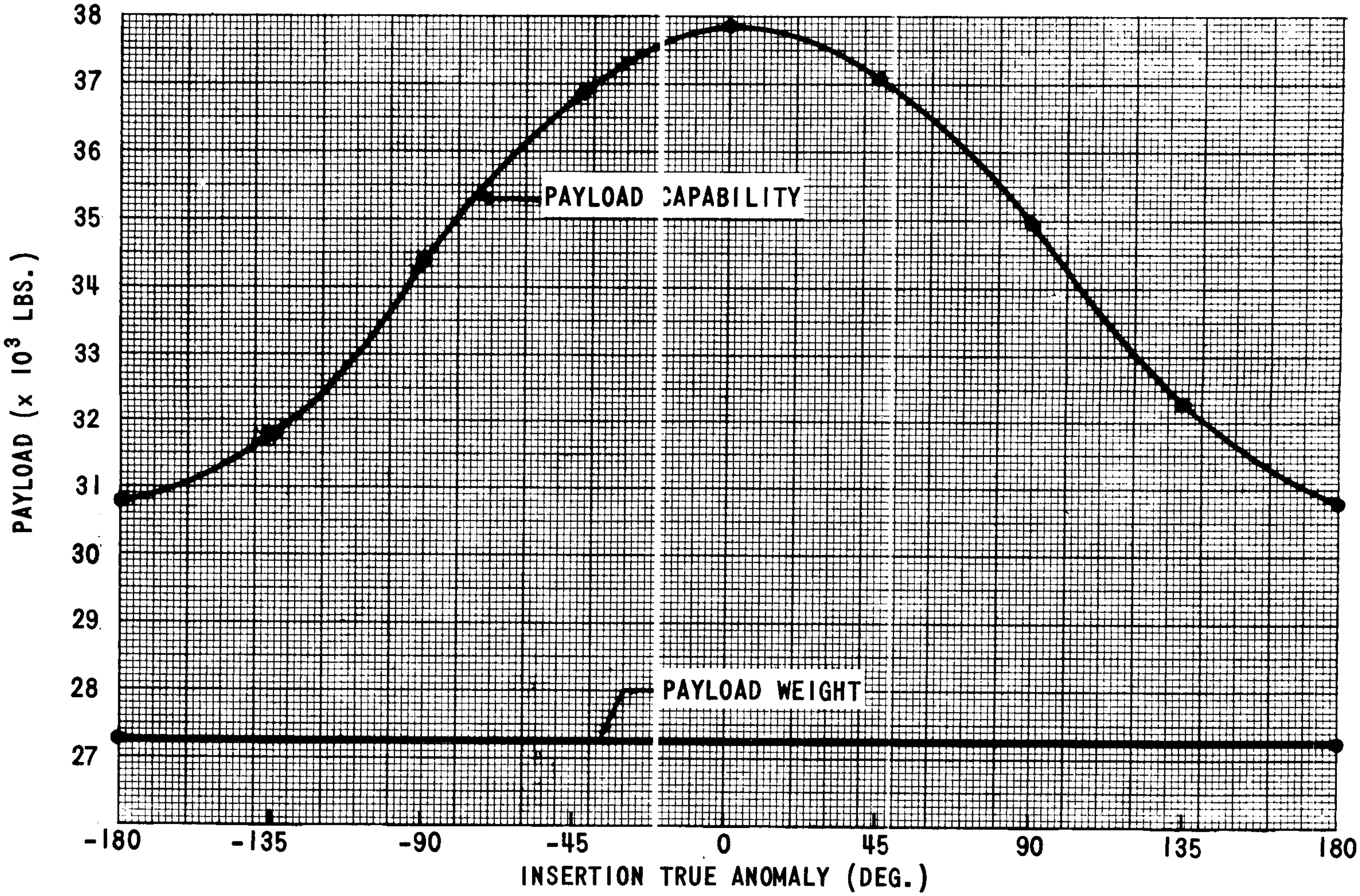


FIGURE 3 - PAYLOAD CAPABILITY FOR THE MSML: SLA NOSECONE JETTISONED

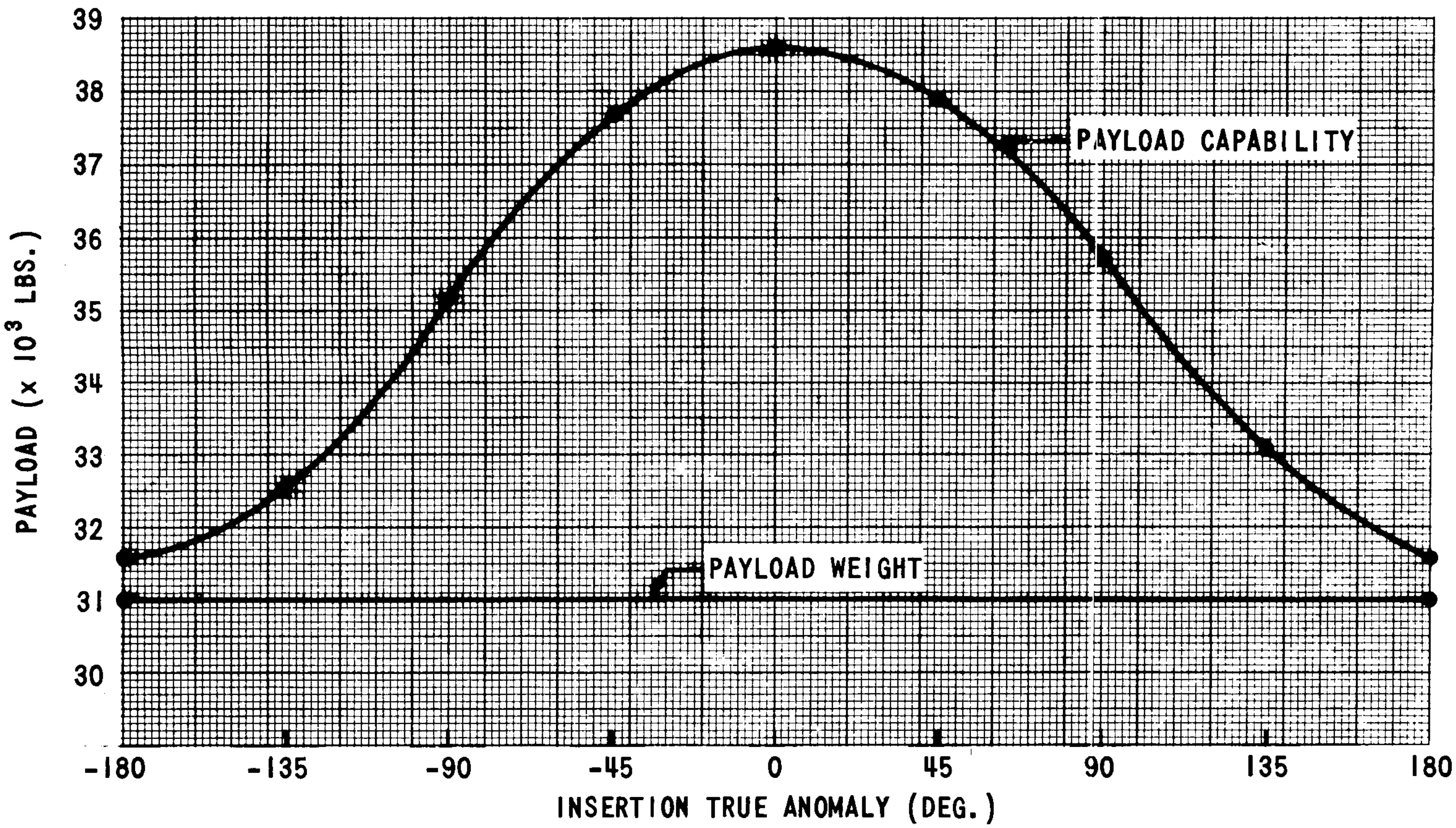


FIGURE 4 - PAYLOAD CAPABILITY FOR THE MSML: SLA NOSECONE NOT JETTISONED

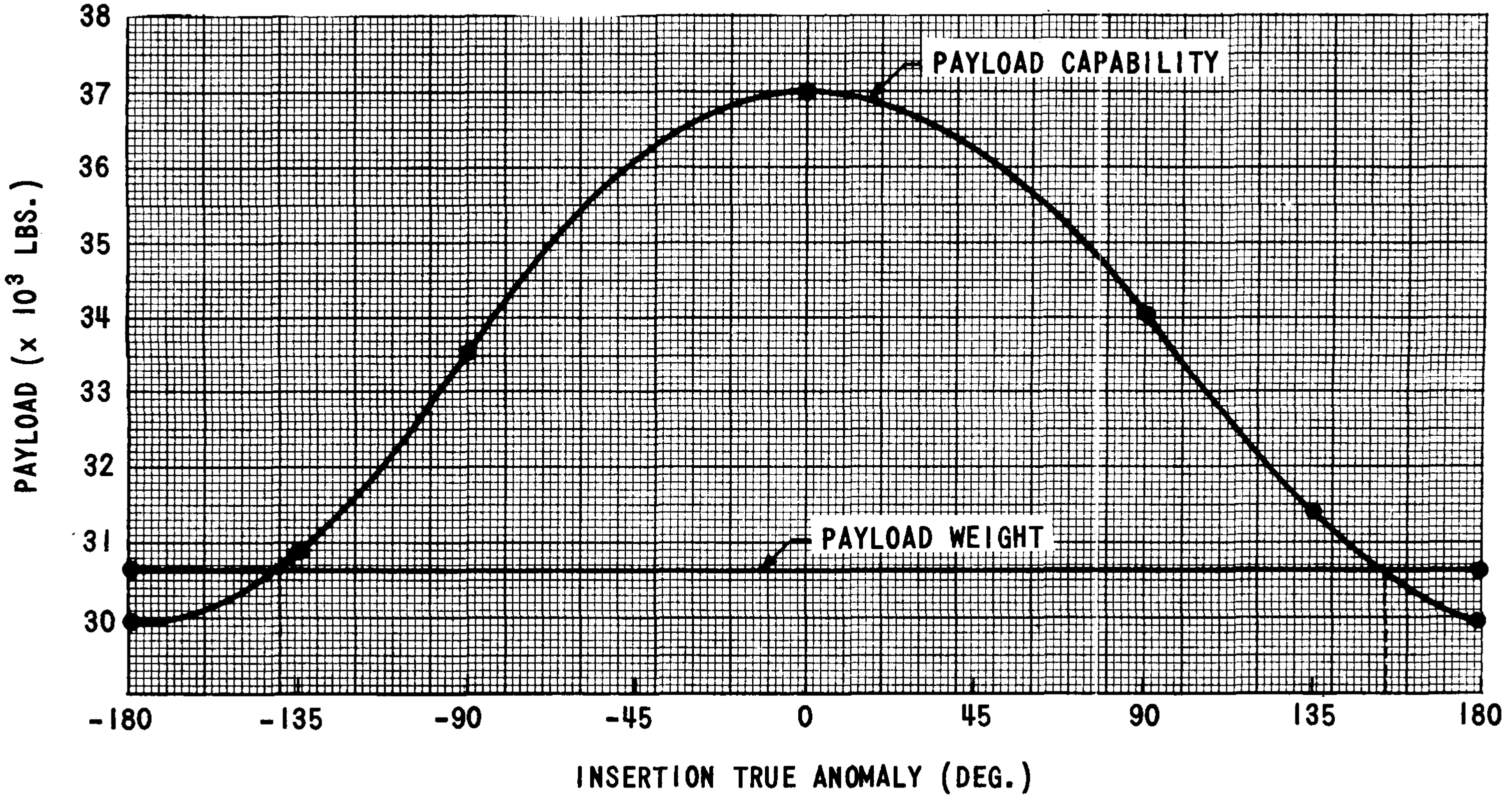
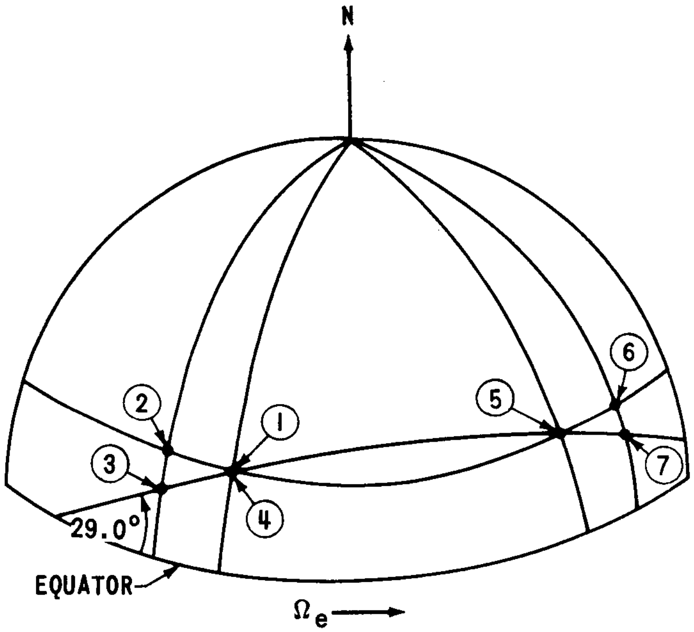


FIGURE 5 - PAYLOAD CAPABILITY FOR THE CM/SM



LEGEND:

1. POSITION OF THE CSM AND MSML LAUNCH SITE ON DAY 0
2. POSITION OF THE CSM LAUNCH SITE AT THE OPENING OF THE LAUNCH WINDOW - DAYS 1 THRU 5
3. INTERSECTION OF THE TARGET ORBIT WITH THE CSM LAUNCH SITE MERIDIAN AT THE OPENING OF THE LAUNCH WINDOW - DAYS 1 THRU 5
4. FIRST COPLANAR CROSSING
5. SECOND COPLANAR CROSSING
6. POSITION OF THE CSM LAUNCH SITE AT THE CLOSING OF THE LAUNCH WINDOW
7. INTERSECTION OF THE TARGET ORBIT WITH THE CSM LAUNCH SITE MERIDIAN AT THE CLOSING OF THE LAUNCH WINDOW - DAYS 0 THRU 5

FIGURE 6 - RELATIVE POSITIONS OF THE CSM LAUNCH SITE AND MSML TARGET ORBIT

TABLE I

Case Number	Orbit Parameters (nm) (Apogee/Perigee)	Prime Deorbit Δv (fps)	SPS Propellant (lbs)	CSM Injected Weight (lbs)	Backup Deorbit Δv (fps)	RCS Deorbit Propellant (lbs)
1	170/170	382	985	26427	251	745
2	200/150	435	1124	26566	216	644
3	220/140	470	1216	26658	198	594
4	240/130	504	1308	26752	180	540
5	275/120	563	1465	26907	162	491

TABLE II

Case Number	Assumed Non-Deorbit Mission Limit (lbs)	RCS Backup Deorbit Propellant (lbs)	Subtotal (lbs)	Gaging (6%) (lbs)	Total (lbs)	Margin (lbs)
1	600	744	1344	80	1424	-146
2	600	645	1245	74	1316	-38
3	600	594	1194	71	1265	+13
4	600	540	1140	68	1208	+70
5	600	492	1092	65	1157	+121

TABLE III

RCS PROPELLANT REQUIREMENTS (MSML MISSION)

Maneuver	Weight* (lbs)	Δv (fps)	RCS Propellant (lbs)		Comments
			Minimum	Most Likely**	
1.0 Prelaunch thru Hard Dock					The laboratory is in an orbit of 130 x 240 nm.
1.1 Prelaunch	26,752†		5.2	7.0	
1.2 CSM Injection	26,745				The CSM is injected into a 180 x 240 nm orbit 100 nm behind the laboratory (worst case down range dispersion).
1.3 CSM Separation	26,745	5.0	15.0	15.0	
1.4 Hybrid Phasing	26,730	15.0	46.0	115.0	The phasing orbit, 121.63 x 240 nm, is based on an 80 nm LAG to the TPI point and a 2 orbit catchup.
1.5 Corrective Combination	26,617	5.0	15.0	45.0	
1.6 TPI	26,572	7.5	23.0	23.0	TPI occurs 20 nm behind the laboratory. The resultant orbit is 125.81 x 240 nm with a 360° transfer to TPF.
1.7 Mid-Course No. 1			0	44.0	
1.8 Mid-Course No. 2			0	44.0	
1.9 TPF	26,461	7.5	23.0	23.0	
1.10 Station Keeping	26,438	10.0	30.0	60.0	
1.11 Hard Dock	26,378		30.0	45.0	

*The weight given is the weight prior to the maneuver.

**Most likely values are based on the ratios of estimated vs actual propellant for the various maneuvers in the Gemini Program.

†The initial weight includes 1310 lbs of SPS prime deorbit propellants.

TABLE III

RCS PROPELLANT REQUIREMENTS (MSML MISSION) (Contd.)

Maneuver	Weight* (lbs)	Δv (fps)	RCS Propellant (lbs)		Comments
			Minimum	Most Likely**	
2.0 Other RCS Requirements					
2.1 IMU-Alignments and Attitude Hold (pre-dock and after CSM/MSML separation)			21.0	42.0	
2.2 CSM/MSML Orbital Operations			25.0	75.0	
2.3 CSM Reentry Operations			47.0	51.0	
Subtotals		50.0	280.2	589.0	
RCS Backup Deorbit				540.0	Δv = 179.2 fps
Total				1129.0	
Gaging (6% of Total)				68.0	
Total RCS Propellants Required				1197.0	
Total RCS Propellants Available				1278.0	
Usable Margin for Contingency				+ 81.0	

TABLE IV

PRELIMINARY CM/SM RENDEZVOUS TIMELINE (-3σ DOWN RANGE INSERTION)

Maneuver	Ground Elapsed Time			Phase Time			Comments
	Hr	Min	Sec	Hr	Min	Sec	
				0	11	0	
(1) CM/SM Injection	0	11	0				
				0	10	0	
(2) CM/SM Separation	0	21	0				
				1	21	23	
(3) Hybrid Phasing	1	42	23				
				1	31	14	2 orbits for phasing
(4) Corrective Combination	3	13	37				
				1	31	14	
(5) TPI	4	44	51				
				1	31	19	360 degree transfer
(6) TPF	6	16	10				
				0	45	39	Station keeping required for 1/2 orbit to meet lighting constraints
(7) Station Keeping Complete	7	1	49				
				0	15	0	
(8) Hard Dock Complete	7	16	49				

TABLE V

Day Number	Time of Launch Window Opening*			Time of Launch Window Closing*			General Range (deg)	Specific Range for $W_T = 17.53$ (deg)
	hr	min	sec	hr	min	sec		
0	0	0	0	1	48	22	$[W_T, W_T + 26.76]$	[18, 44]
1	23	12	14	25	19	51	$[W_T - 15.68, W_T + 15.67]$	[2, 33]
2	46	43	48	48	51	25	$[W_T - 27.34, W_T + 4.18]$	[-10, 22]
3	70	15	22	72	22	59	$[W_T - 38.83, W_T - 7.31]$	[-21, 10]
4	93	46	48	95	54	25	$[W_T - 50.32, W_T - 18.80]$	[-33, -1]
5	117	18	36	119	26	13	$[W_T - 61.81, W_T - 30.29]$	[-44, -13]

*The times of launch window opening and closing are from MSML launch

BELLCOMM, INC.

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From: W. L. Austin

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